NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE

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INVESTIGATION OF BOUNDARY-LAYER REYNOLDS NUMBER FOR TRANSITION ON AN NACA $65_{(215)}$-114 AIRFOIL IN THE LANGLEY TWO-DIMENSIONAL LOW-TURBULENCE PRESSURE TUNNEL

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SUMMARY

A low-turbulence wind-tunnel investigation was made of an aerodynamically smooth NACA $65_{(215)}$-114 airfoil having faired surfaces back to 37 percent chord to determine the magnitude of the boundary-layer Reynolds number at various positions of transition from laminar to turbulent flow along both airfoil surfaces. In addition to boundary-layer measurements, values of the section drag coefficient were obtained by means of the wake-survey method.

The boundary-layer Reynolds number $\left(R_{\delta_{cr}}\right)$ was found to vary in magnitude from approximately 6700 to 8000 at positions of transition ranging from 50 percent chord to 25 percent chord; the values of $R_{\delta_{cr}}$ were based on the boundary-layer thickness δ, which is defined as the distance from the airfoil surface to a point within the boundary layer where the velocity is equal to 0.707 of the velocity at the outer edge of the boundary layer. The results indicated, however, that for a smooth and faired low-drag-type airfoil operating in the low-drag range in an air stream of low turbulence, transition points and drag coefficients may be estimated within approximately 7 percent chord and 0.0003, respectively, of the actual values by assuming a constant value of $R_{\delta_{cr}}$ of 8000.

INTRODUCTION

In the absence of direct test data, it is sometimes desirable to be able to estimate the positions of transition from laminar to turbulent flow in order to calculate the profile-drag coefficients of airfoil sections. Transition has been estimated to occur in the favorable pressure gradient on smooth low-drag-type airfoils under conditions of low turbulence at values of boundary-layer Reynolds number $R_{\delta_{cr}}$ between 7500 and 9000. These values of $R_{\delta_{cr}}$ correspond to the range of values of R_{δ} measured in a series of flight tests of reference 1. Inasmuch as there appears to be only a limited amount of data on the boundary-layer Reynolds number at which transition occurs, the present investigation was made to obtain additional information on the values of $R_{\delta_{cr}}$ at various positions of transition along airfoil surfaces.

The investigation was made of an NACA $65_{(215)}$-114 airfoil in the Langley two-dimensional low-turbulence pressure tunnel. Boundary-layer surveys were made at several stations on the upper and lower surfaces of the airfoil model through a range of free-stream Reynolds number up to approximately 58.0×10^6. In addition, profile-drag coefficients were measured by means of the wake-survey method through a range of free-stream Reynolds number up to 40.0×10^6.

SYMBOLS

c	airfoil chord
c_l	section lift coefficient
c_d	section drag coefficient
ρ_o	free-stream density
U_o	free-stream velocity
q_o	free-stream dynamic pressure $\left(\frac{1}{2}\rho_o U_o{}^2\right)$
H_o	free-stream total pressure
p	static pressure on airfoil surface
h	total pressure inside boundary layer
u	local velocity inside boundary layer $\left(\sqrt{\frac{2(h - p)}{\rho_o}}\right)$
U	local velocity at outer edge of boundary layer
h_1	total pressure measured by tube in contact with surface
S	pressure coefficient $\left(\frac{H_o - p}{q_o}\right)$
x	distance along airfoil chord from leading edge

s	distance along airfoil surface from leading edge
y	distance perpendicular to airfoil surface
y'	distance perpendicular to airfoil chord
ν	coefficient of kinematic viscosity
δ	boundary-layer thickness, distance from airfoil surface to point within boundary layer where velocity is equal to 0.707 velocity at outer edge of boundary layer
R_δ	boundary-layer Reynolds number based on effective boundary-layer thickness $(U\delta/\nu)$
R_o	free-stream Reynolds number based on airfoil chord $(U_o c/\nu)$
R_o'	free-stream Reynolds number based on airfoil chord (uncorrected for tunnel-wall effects)

Subscript:

cr	value at which transition occurs

MODEL AND APPARATUS

Photographs of the model, which was built to the ordinates of the NACA $65_{(215)}$-114 airfoil (table I), are shown in figure 1. A detailed description of the construction of the model, which has an 85-inch chord and 36-inch span, is given in reference 2. The model was glazed to a faired contour back to 37 percent chord, at which station a spar introduced waviness on both surfaces. An indication of the magnitude of these waves is presented in reference 2. Both airfoil surfaces were sanded to aerodynamic smoothness.

Tests were made in the Langley two-dimensional low-turbulence pressure tunnel with the model mounted so that it completely spanned the 3-foot test section. The turbulence level of the tunnel is only a few hundredths of 1 percent. A detailed description of the Langley two-dimensional low-turbulence pressure tunnel is presented in reference 3. A multitube pressure "mouse", described in reference 4, was used in obtaining the boundary-layer surveys and the pressure distributions over the airfoil. The heights of the total-pressure tubes above the airfoil surface were measured with a micrometer microscope.

TEST METHODS AND TESTS

Drag measurements were made by the wake-survey method and reduced to free-air conditions as described in reference 3. The boundary-layer velocity distributions were obtained by measuring the static pressure outside the boundary layer and the total pressure at several positions within the boundary layer with the multitube mouse. The airfoil pressure distribution was obtained from the readings of the static-pressure tube on the mouse. At each station, the static-pressure tube was bent approximately to the airfoil contour at about 1/4 inch from the airfoil surface. One total-pressure tube was bent so that it remained on the airfoil surface regardless of the air loads imposed on the mouse.

The method used in determining the free-stream Reynolds number at which transition occurs at a given station is similar to that used in reference 4. A parameter $\frac{\sqrt[3]{h_1 - p}}{\sqrt{H_0 - p}}$ was used which remained substantially constant while the flow in the boundary layer was laminar and which increased in value as the flow became turbulent. This parameter was plotted against the uncorrected free-stream Reynolds number R_0', and transition was taken as the point corresponding to the knee of the curve. In those cases where the knee of the curve was not sharply defined, the shapes of the boundary-layer velocity profiles through a small range of Reynolds number at the knee were used as an aid in determining the value of the Reynolds number for transition.

Drag data were obtained at a section lift coefficient of 0.14 for a range of free-stream Reynolds number up to 40.0×10^6. Boundary-layer and transition measurements, which were made at the same section lift coefficient and at the center line of the model, were obtained for a range of Reynolds number up to approximately 58.0×10^6. By varying the tunnel stagnation pressure from 14.7 pounds to 135 pounds per square inch absolute, it was possible to maintain the tunnel Mach number below 0.2 for the complete range of Reynolds number investigated.

RESULTS AND DISCUSSION

Boundary-layer velocity profiles.- A few representative distributions of velocity through the boundary layer are presented in figure 2 for two stations on the upper airfoil surface. The change in velocity profile from the laminar to the turbulent type with increasing Reynolds number can be seen. Although the chordwise position of minimum pressure on both airfoil surfaces occurs at approximately 46 percent chord (fig. 3), laminar flow was obtained at least as far back as 50 percent chord at the lower values of the Reynolds number.

Transition.– The variations of the transition parameter $\frac{\sqrt[3]{h_1 - p}}{\sqrt{H_o - p}}$ with uncorrected free-stream Reynolds number R_o' are presented in figure 4. The Reynolds number at which transition is considered to occur for each station at which measurements were made is indicated by an arrow. The forward movement of transition with increasing values of the uncorrected Reynolds number is shown for both airfoil surfaces in figure 5. By use of these positions of transition, the corresponding uncorrected Reynolds numbers, and the measured variations of velocity over the airfoil (fig. 3), values of the boundary-layer Reynolds number for transition $\left(\text{critical boundary-layer Reynolds number } R_{\delta_{cr}}\right)$ were calculated by means of the following equation obtained from reference 5:

$$\frac{{R_{\delta_{cr}}}^2}{R_o'} = (2.3)^2\left(\frac{U_o}{U}\right)_x^{7.17}\int_0^{s/c}\left(\frac{U}{U_o}\right)^{8.17} d\frac{s}{c} \qquad (1)$$

The measured values of boundary-layer thickness were not used for determination of $R_{\delta_{cr}}$ inasmuch as the measured boundary-layer velocity profiles at large values of the Reynolds number were considered to be too unreliable because of deflections of the total-head tubes at high values of air-stream dynamic pressure.

Figure 6, which presents the values of $R_{\delta_{cr}}$ plotted against the position of transition, indicates that $R_{\delta_{cr}}$ varies from approximately 8000 to 7250 at positions of transition along the airfoil chord ranging from 25 percent chord to 37 percent chord. The value of $R_{\delta_{cr}}$ decreased in magnitude at positions of transition behind 37 percent chord, reaching a minimum of approximately 6700 at 50 percent chord. This decrease in the value of $R_{\delta_{cr}}$ may have been partially caused by the surface waviness at the spar located at 37 percent chord.

Values of $R_{\delta_{cr}}$ from 7400 to 9200 were obtained on the upper surface of an NACA 35-215 airfoil in flight (reference 1). In reference 1, however, it is stated that individual values measured during that investigation may not be entirely reliable but that the results are sufficiently consistent to indicate attainment of values of R_δ of approximately 8000. Although the pressure distribution of the NACA 35-215 airfoil is more favorable than the pressure distribution of the NACA $65_{(215)}$-114 airfoil, approximately the same values of $R_{\delta_{cr}}$ were measured for the two models. Disturbing influences, such as surface roughness, air-stream turbulence, and possibly vibration, however, are known to have large effects on the position of transition and the corresponding values of $R_{\delta_{cr}}$, whereas the models investigated had surfaces of aerodynamic smoothness and were tested in air streams of low turbulence.

Effect of constant $R_{\delta_{cr}}$ concept on transition and drag.- The value of boundary-layer Reynolds number for transition has been shown in figure 6 to vary in magnitude for positions of transition ranging from 50 percent chord to 25 percent chord on both airfoil surfaces. In order to determine the accuracy with which the positions of transition and corresponding drag coefficient can be approximated by means of assuming a constant value of $R_{\delta_{cr}}$ in conjunction with the theoretical airfoil pressure distribution, transition points were calculated by use of equation (1) and drag coefficients by use of the method of reference 6 with the theoretical pressure distribution at the test lift coefficient of 0.14 and constant values of $R_{\delta_{cr}}$ of 7500 and 8000.

The variations of the estimated transition points with Reynolds number are presented in figure 7. In order to provide a basis of comparison for the positions of transition calculated by use of a constant value of $R_{\delta_{cr}}$ and the theoretical pressure distribution, the variation of the actual positions of transition with Reynolds number under free-air conditions is presented. These curves were obtained by means of equation (1) after correcting the measured airfoil pressure distribution for the effects of the tunnel walls (see fig. 3 and reference 7) and by assuming that at any given station along the airfoil surfaces the critical boundary-layer Reynolds number would be the same in free air as that measured in the wind tunnel. Figure 7 indicates that use of a constant value of $R_{\delta_{cr}}$ of 7500 or 8000 and the theoretical pressure distribution results in estimates of the transition point that may be in error by no more than 7 percent chord at Reynolds numbers ranging from 26.0×10^6 to 49.0×10^6. The largest discrepancy of 7 percent chord was noted for the upper surface at a Reynolds number of 26.0×10^6 where transition occurred at 41 percent chord. At those Reynolds numbers at which transition occurred behind 37 percent chord, the surface waviness at the model spar possibly caused transition to occur slightly forward of the normal position for a completely faired airfoil.

Section drag coefficients, calculated by the use of the estimated positions of transition and the theoretical pressure distribution in accordance with the method of reference 6, are compared in figure 8 with section drag coefficients measured by means of the wake-survey method and reduced to free-air conditions as described in reference 3. Values of section drag coefficient calculated by use of a constant $R_{\delta_{cr}}$ of 8000 and the theoretical pressure distribution are within 0.0003 of the drag coefficients obtained by the wake-survey method up to a Reynolds number of 40.0×10^6. Although wake-survey measurements were not made at larger values of the Reynolds number, comparison of the estimated transition points with measured transition points presented in figure 7 indicates that the use of a constant $R_{\delta_{cr}}$ of 8000 will result in calculated drag coefficients within 0.0003 of the actual values at Reynolds numbers as large as at least 55.0×10^6.

CONCLUDING REMARKS

A low-turbulence wind-tunnel investigation was made of an aerodynamically smooth NACA $65_{(215)}$-114 airfoil having faired surfaces back to 37 percent chord. Values of boundary-layer Reynolds numbers at which transition was observed $\left(R_{\delta_{cr}}\right)$ varied from 6700 to 8000 at positions of transition ranging from 50 percent chord to 25 percent chord. The results indicated that for a smooth and faired low-drag-type airfoil operating in the low-drag range in an air stream of low turbulence, the use of a fixed value of $R_{\delta_{cr}}$ of 8000 yields estimates of the transition points and drag coefficient within approximately 7 percent chord and 0.0003, respectively, of the actual values at Reynolds numbers between the maximum value at which transition occurs at the point of minimum pressure up to a value of at least 55.0×10^6.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., July 8, 1948

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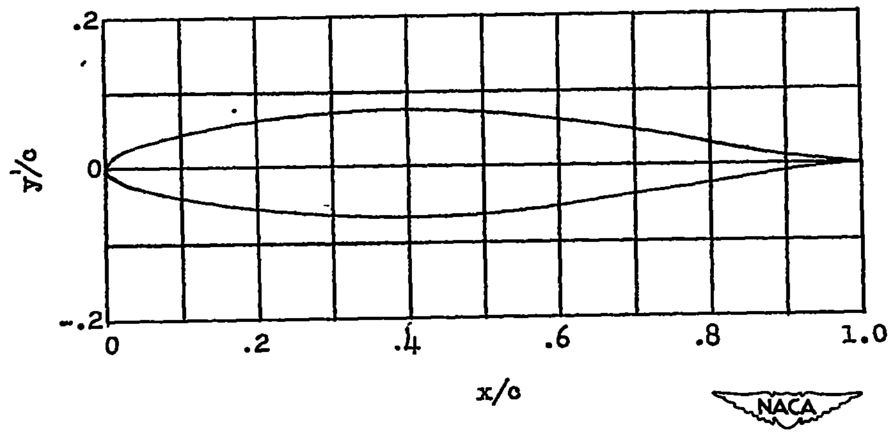
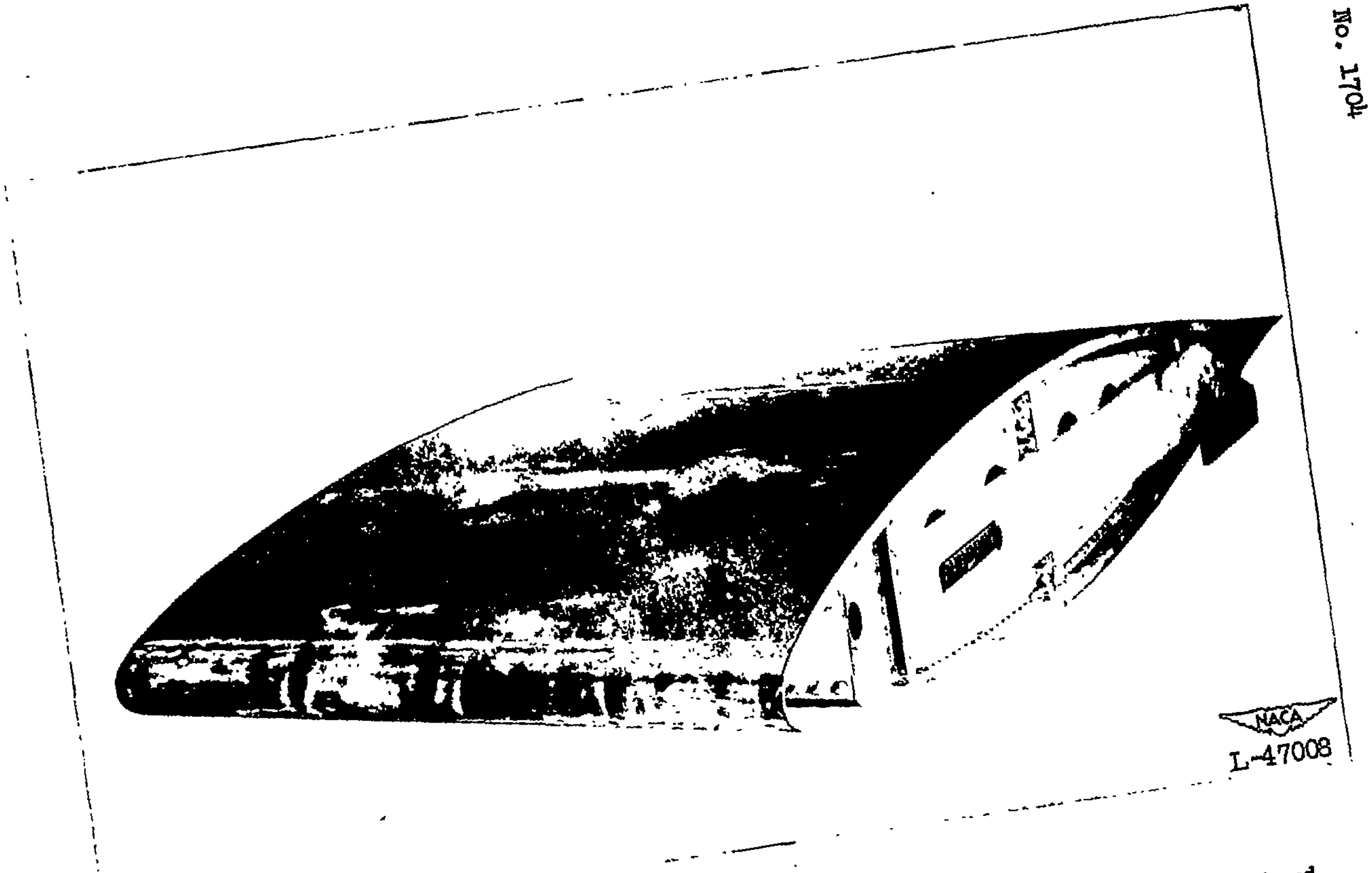


TABLE I

ORDINATES OF THE NACA $65_{(215)}$-114 AIRFOIL

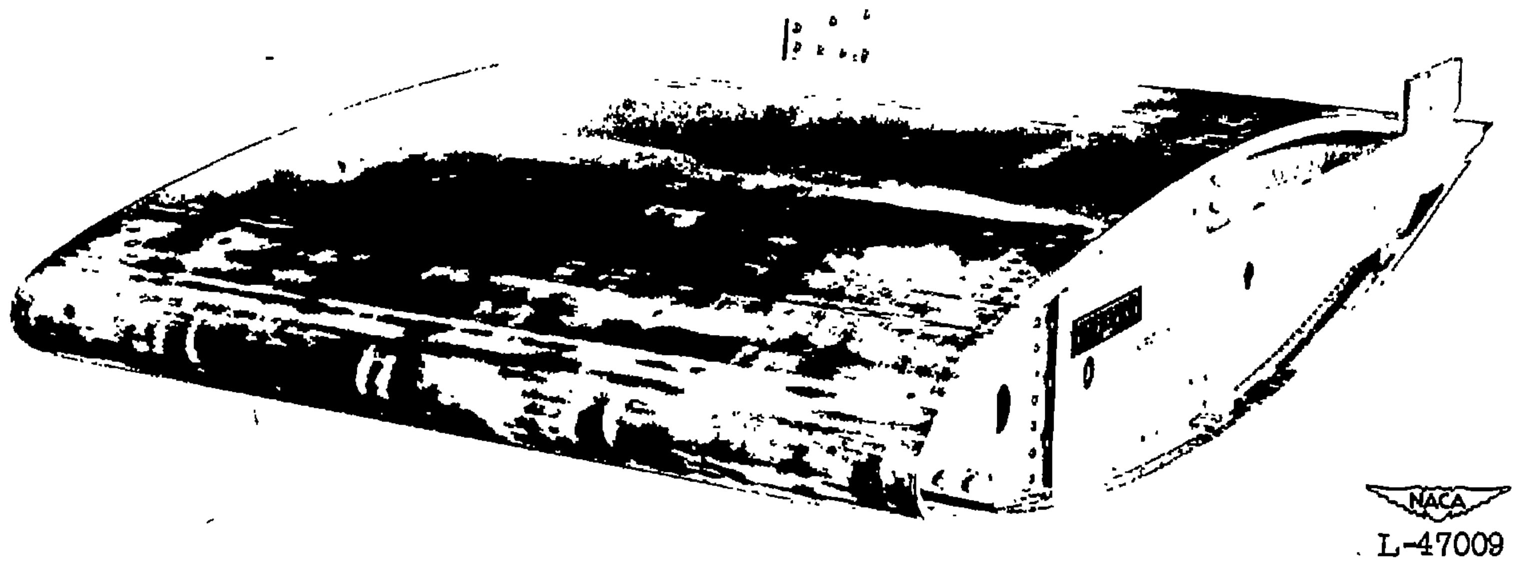
[Stations and ordinates given in percent of airfoil chord]

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.456	1.073	.544	-1.023
.701	1.300	.799	-1.230
1.195	1.642	1.305	-1.534
2.437	2.261	2.563	-2.075
4.929	3.186	5.071	-2.870
7.426	3.906	7.574	-3.482
9.926	4.508	10.074	-3.992
14.929	5.472	15.071	-4.800
19.936	6.206	20.064	-5.410
24.945	6.761	25.055	-5.865
29.955	7.161	30.045	-6.189
34.966	7.418	35.034	-6.388
39.977	7.534	40.023	-6.462
44.989	7.480	45.011	-6.384
50.000	7.242	50.000	-6.138
55.010	6.820	54.990	-5.724
60.018	6.246	59.982	-5.174
65.025	5.558	64.975	-4.528
70.029	4.779	69.971	-3.807
75.031	3.942	74.969	-3.046
80.029	3.065	79.971	-2.269
85.025	2.181	84.975	-1.509
90.019	1.326	89.981	-.810
95.009	.557	94.991	-.241
100.000	0	100.000	0
L.E. radius: 1.311 Slope of radius through L.E.: 0.042			



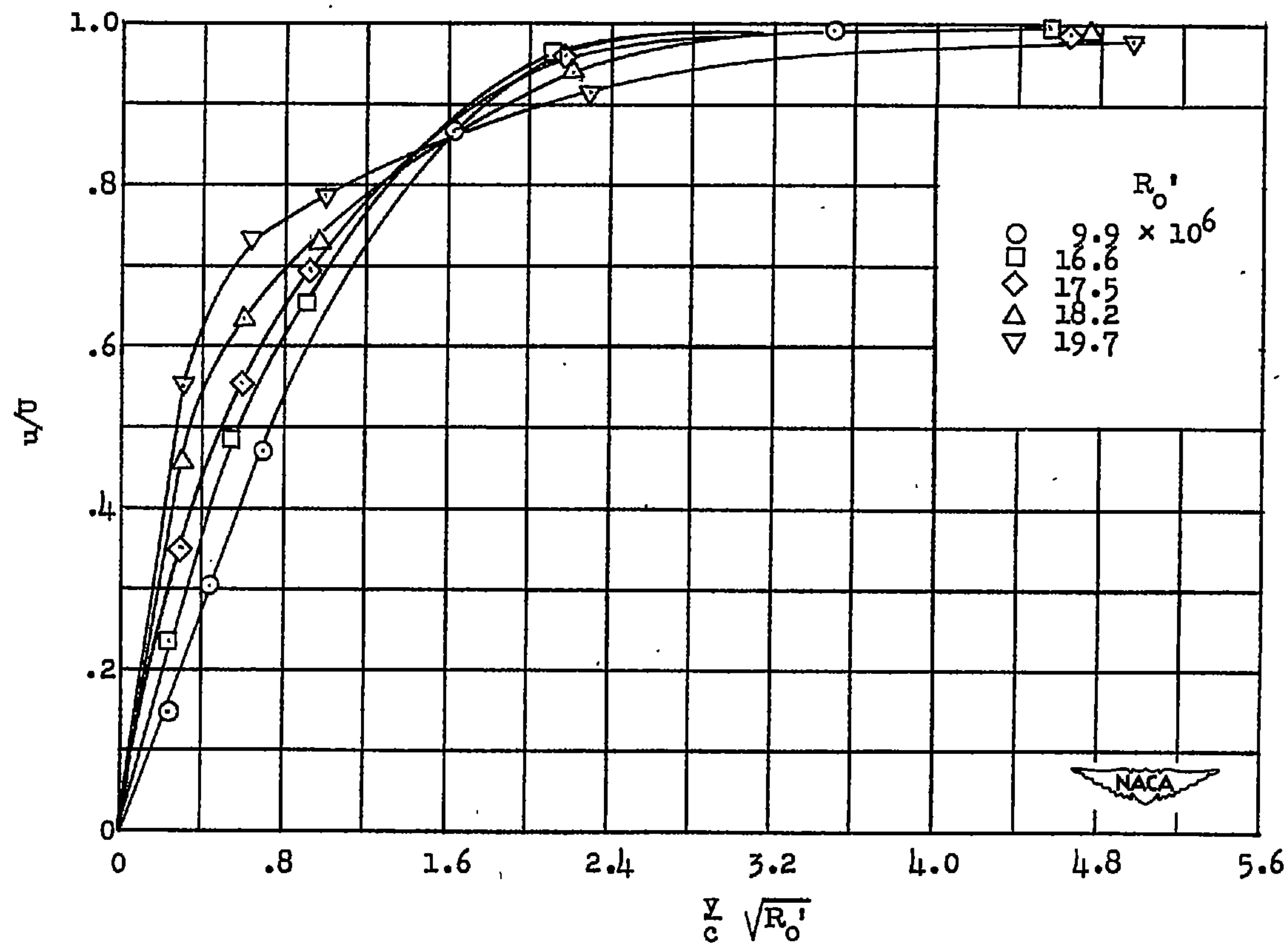
(a) Upper surface.

Figure 1.- NACA $65_{(215)}$-114 airfoil with model surfaces glazed and sanded to 50 percent chord.



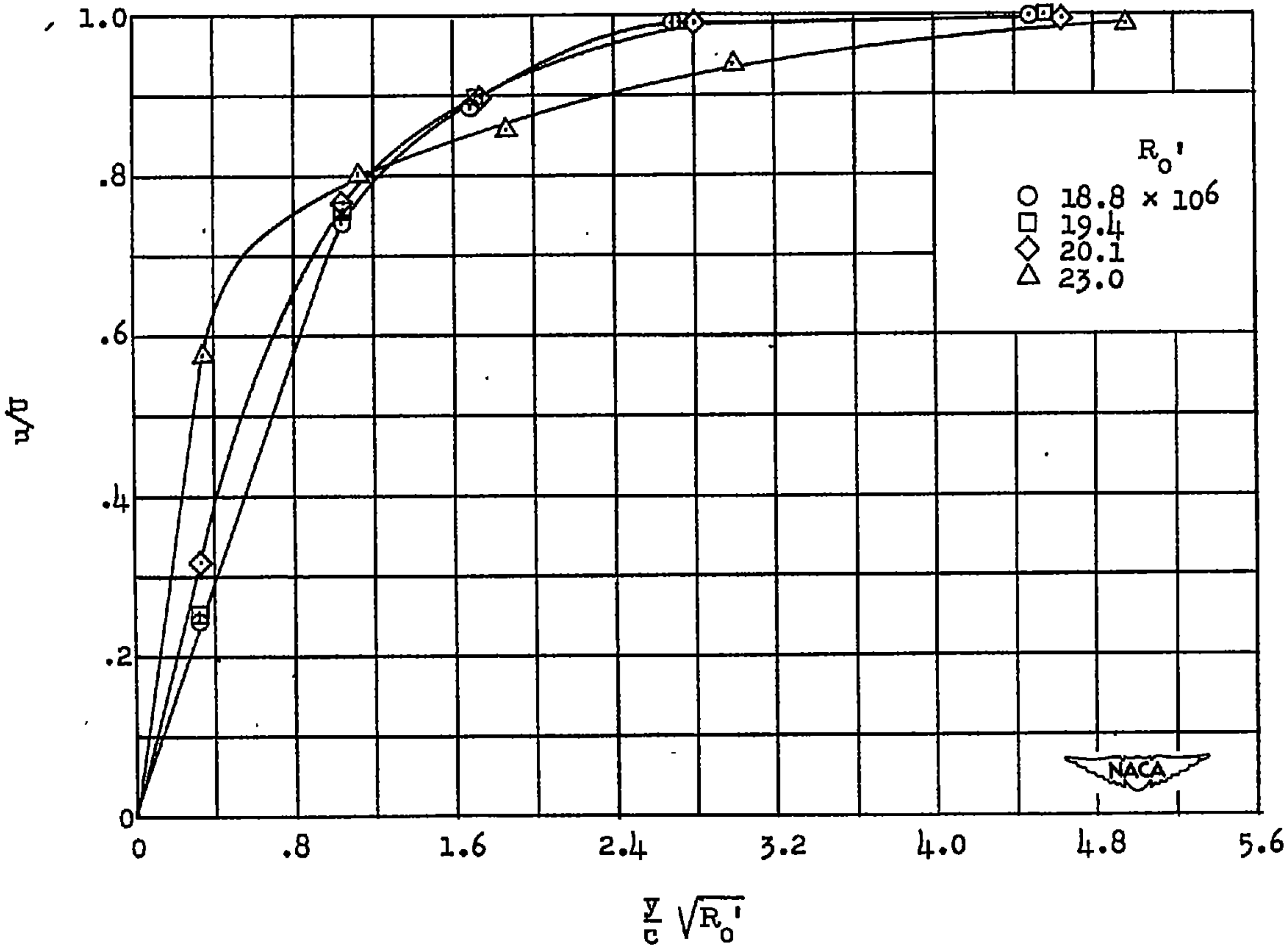
(b) Lower surface.

Figure 1.- Concluded.



(a) $\frac{x}{c} = 0.50.$

Figure 2.- Boundary-layer surveys obtained on upper surface of NACA $65_{(215)}$-114 airfoil.



(b) $\frac{x}{c} = 0.45$.

Figure 2.- Concluded.

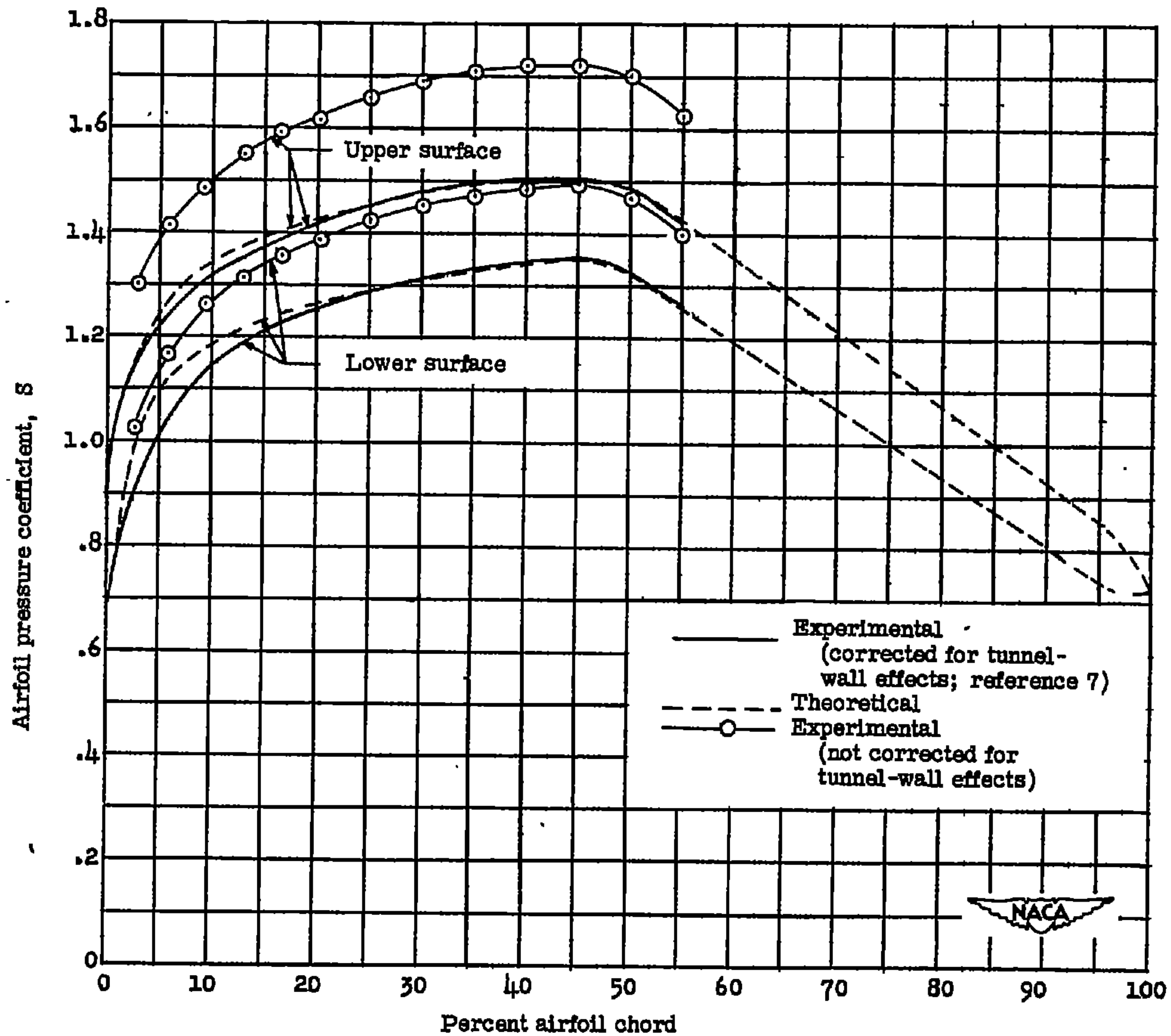


Figure 3.- Pressure distribution for NACA $66_{(215)}$-114 airfoil. $c_l = 0.14$.

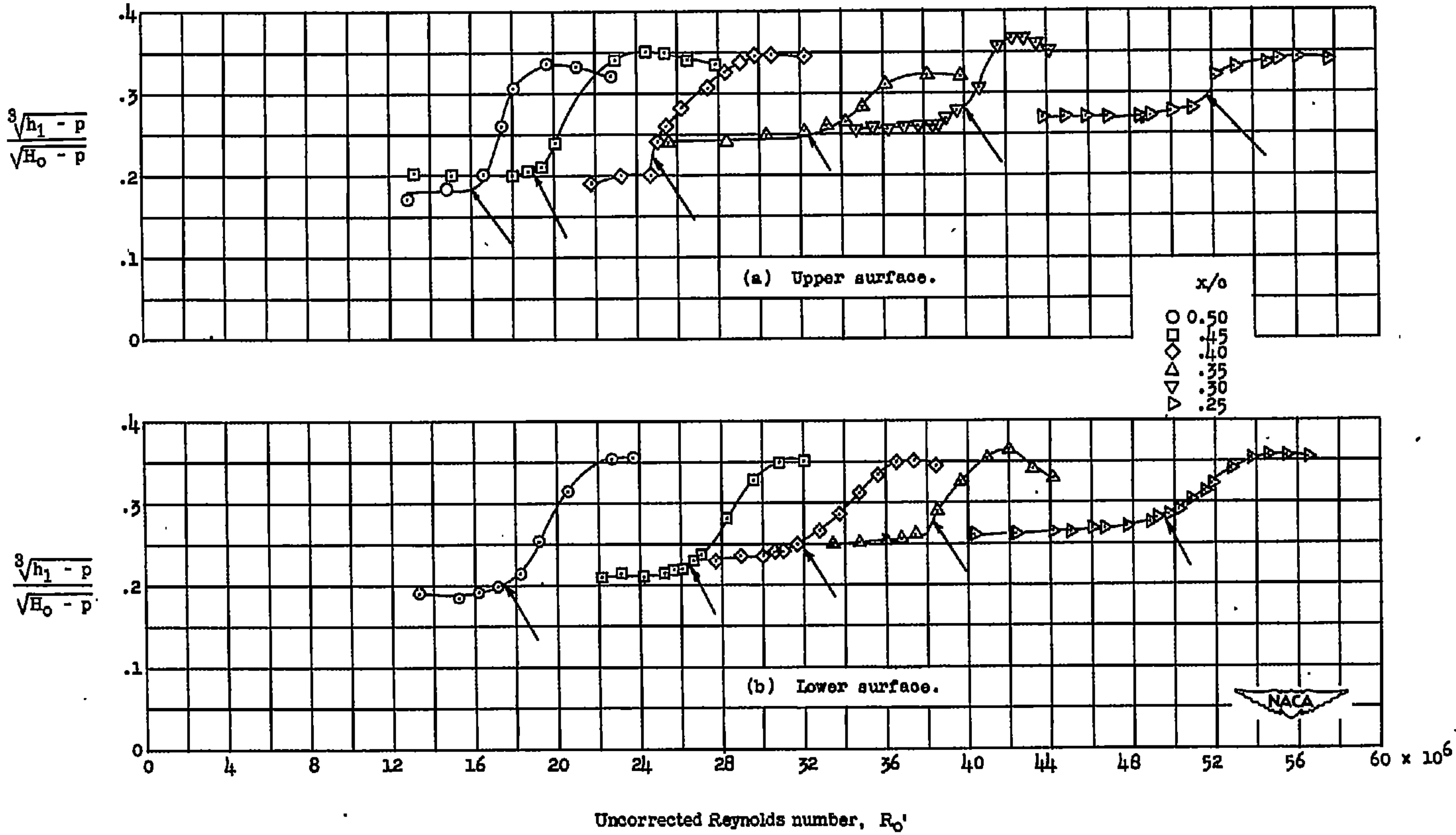


Figure 4.- Transition parameter as a function of uncorrected Reynolds number for NACA $66_{(215)}$-114 airfoil. $c_l = 0.14$.

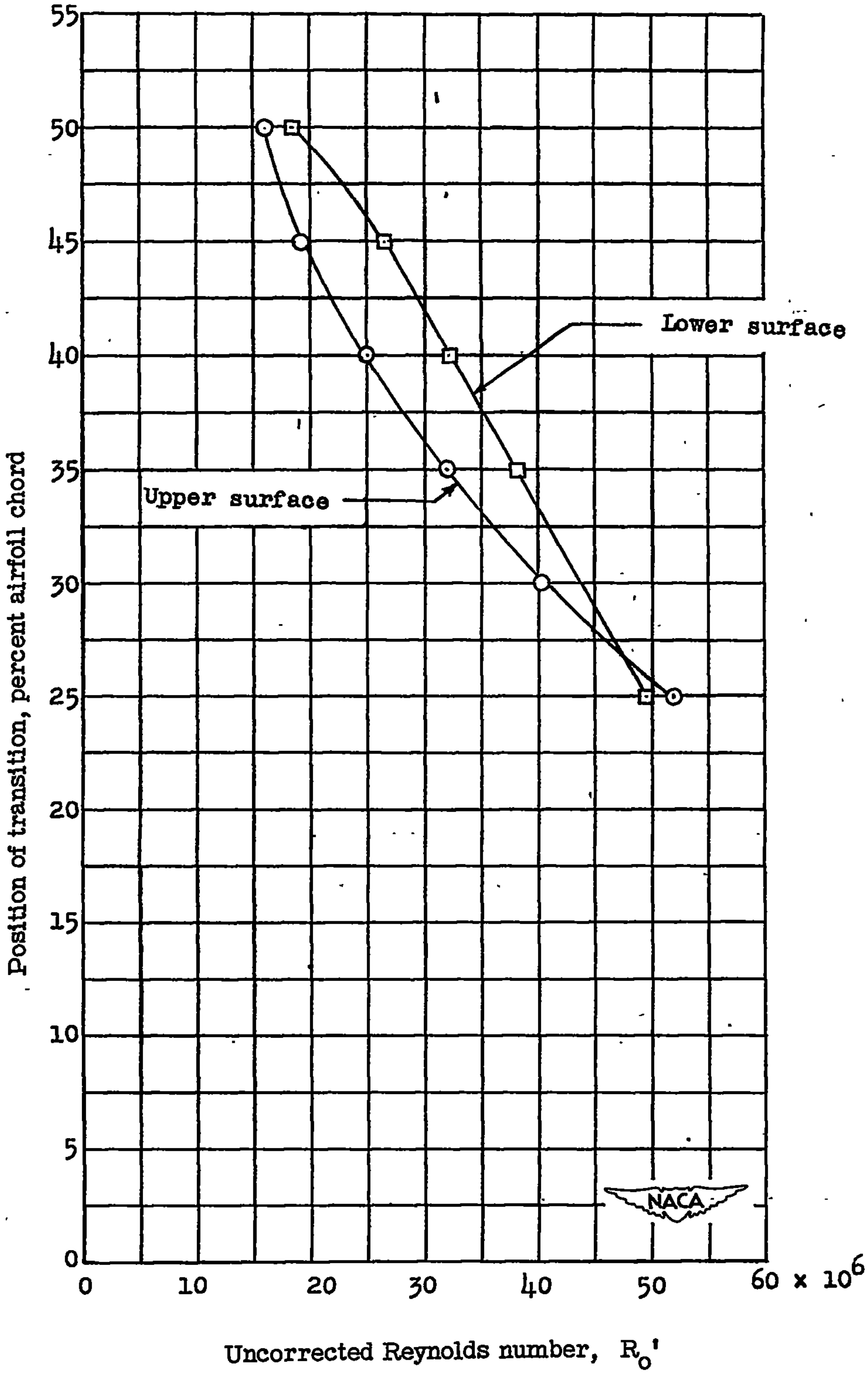


Figure 5.- Variation of position of transition with uncorrected Reynolds number for the NACA $65_{(215)}$-114 airfoil; $c_l = 0.14$.

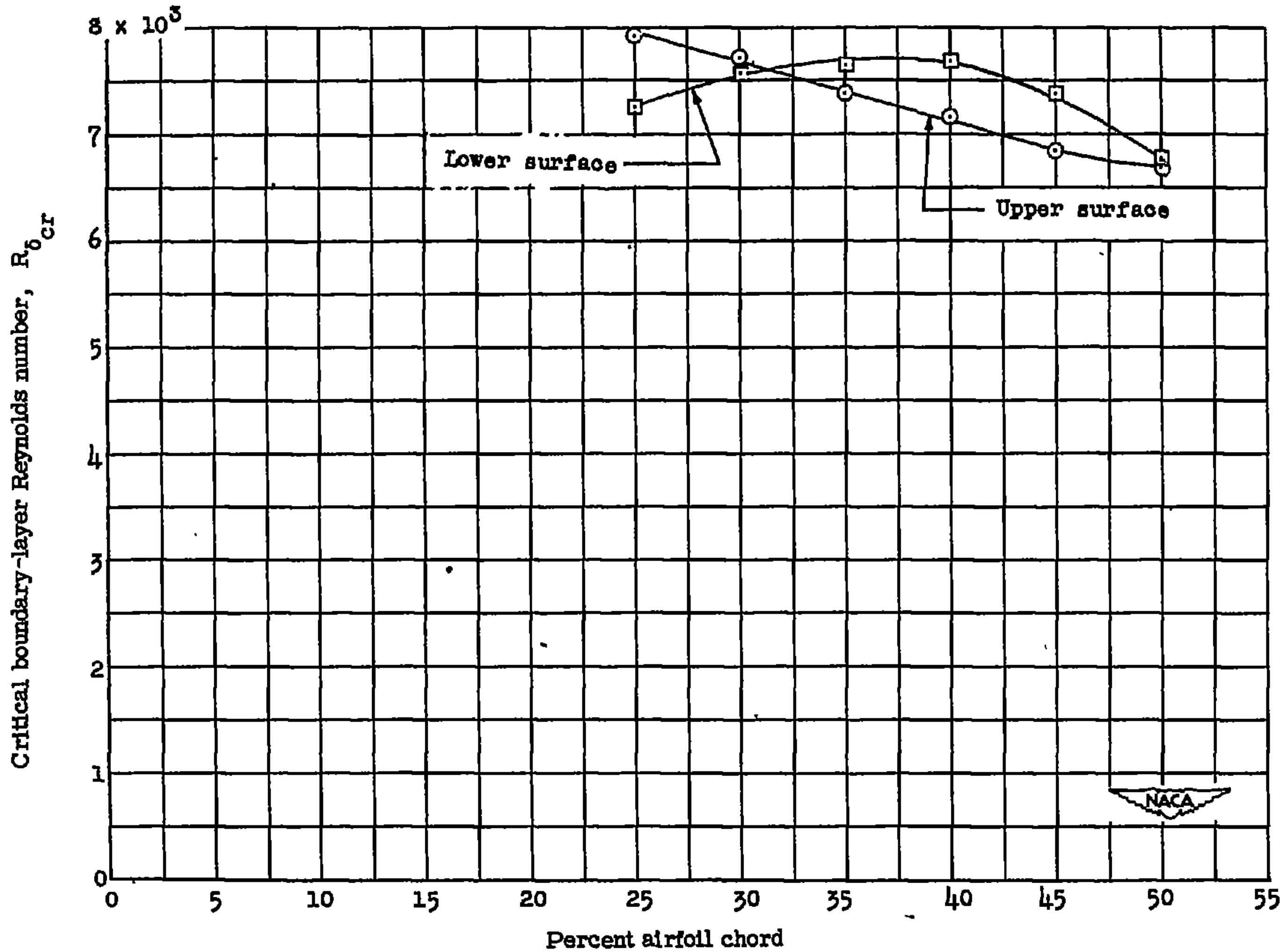


Figure 6.- Variation of critical boundary-layer Reynolds number with airfoil chord for NACA $65_{(215)}$-114 airfoil. $c_l = 0.14$.

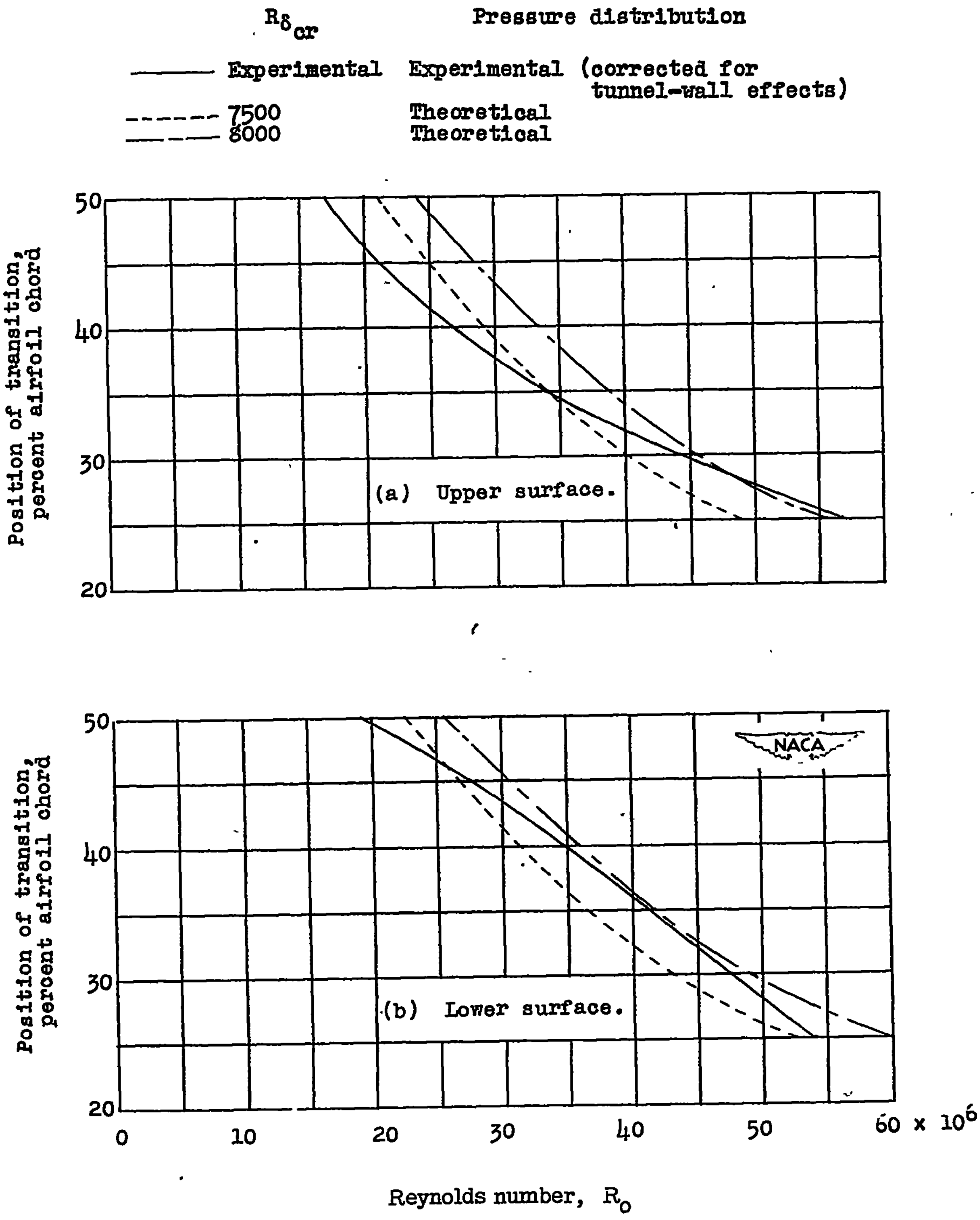


Figure 7.- Variation of position of transition with Reynolds number for several critical values of boundary-layer Reynolds number on NACA $65_{(215)}$-114 airfoil. c_l = 0.14.

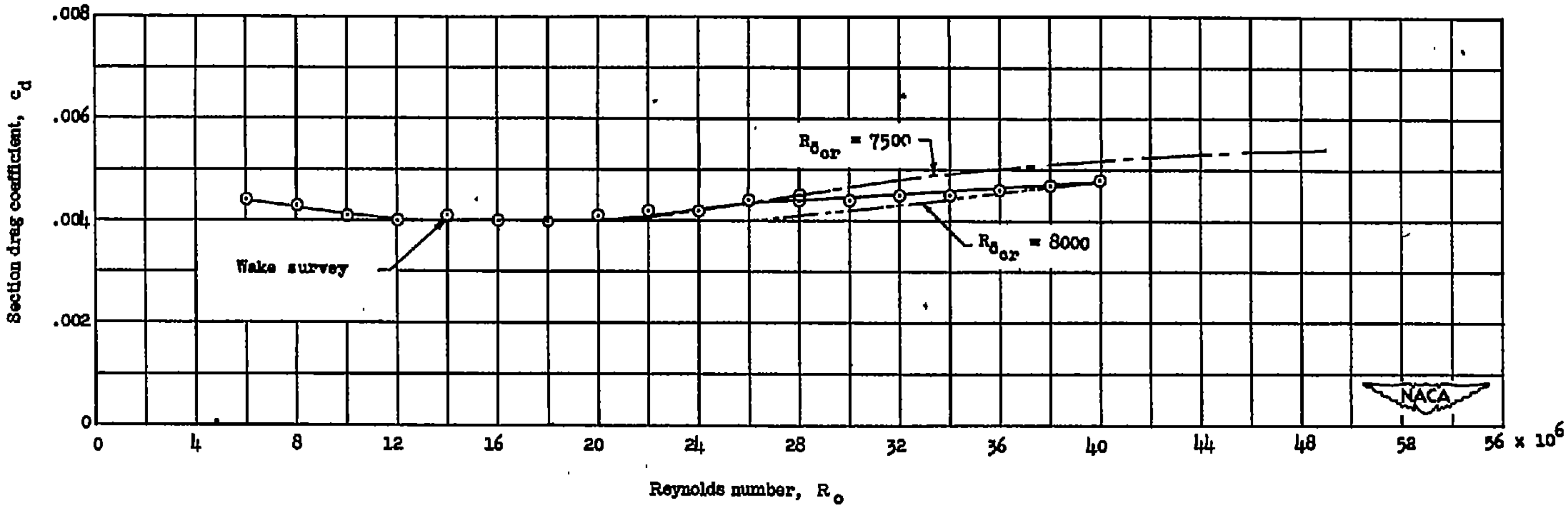


Figure 8.- Variation of section drag coefficient with Reynolds number for NACA $65_{(215)}$-114 airfoil. $c_l = 0.14$.